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HOVERING AND LOW-SPEED PERFORMANCE AND CONTROL CHARACTERISTICS OF AN AERODYNAMIC-SERVOCONTROLLED HELICOPTER ROTOR SYSTEM AS DETERMINED ON THE LANGLEY HELICOPTER TOWER

By Paul J. Carpenter and Russell S. Paulnock

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Langley Air Force Base, Va.

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HOVERING AND LOW-SPEED PERFORMANCE AND CONTROL CHARACTERISTICS OF AN AERODYNAMIC-SERVOCONTROLLED HELICOPTER ROTOR SYSTEM AS DETERMINED ON THE LANGLEY HELICOPTER TOWER

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SUMMARY

An investigation has been conducted with the Langley helicopter tower to obtain basic performance and control characteristics of an aerodynamic-servocontrolled rotor system. Blade-pitch control is obtained in this configuration by utilizing an auxiliary flap to twist the blades. Rotor thrust and power required were measured for the hovering condition and over a range of wind velocities from 0 to 30 miles per hour. The control characteristics and the transient response of the rotor to various control movements were also measured.

The hovering-performance data are presented as a survey of the wake velocities and the variation of torque coefficient with thrust coefficient. The power required for the test rotor to hover at a thrust of 1350 pounds and a rotor speed of 240 rpm is approximately 6.5 percent greater than that estimated for a conventional rotor of the same diameter and solidity. It is believed that most of this difference is caused by the flap servomechanism. The reduction in total power required for sustentation of the configuration tested at various wind velocities and at the normal operating rotor thrust was found to be similar to the theoretical and experimental results for rotors with conventionally actuated pitch. The control effectiveness was determined as a function of rotor speed. Sufficient control was available to give a thrust range of 0 to 1500 pounds and a rotor tilt of $\pm 7^{\circ}$. The time lag between flap motion and blade-pitch response is approximately 0.02 to 0.03 second. The response of the rotor following the blade-pitch response is similar to that of a rotor with conventionally actuated pitch changes.

The over-all characteristics of the rotor investigated indicate that satisfactory performance and control characteristics were obtained.

INTRODUCTION

Some type of servomechanism to control the blade pitch of a helicopter rotor may be desirable in certain cases, for example, in large rotors where excessive control forces and pitching moments may be encountered. Accordingly, tests were made on the Langley helicopter tower to determine the performance and control characteristics of an aerodynamic-servocontrolled helicopter rotor system. It was intended that an investigation of this configuration would give fundamental information of a general nature on rotors with such control systems.

The rotor differs from conventional rotors by the unique method of controlling the blade pitch. In this rotor configuration, the blade is attached rigidly at the root, and pitch change is effected by twisting the blade at an outboard station by means of an aerodynamic flap instead of by rotating the blade at the root.

This paper presents measurements of the rotor performance for hovering and low forward speeds. Measurements of the aerodynamic-flap servocontrol characteristics and the transient response of the rotor to various control movements are also included. The results of the measurements are discussed and, in a few cases, comparisons are made with conventional rotors with pitch actuation accomplished by rotating the blade roots.

SYMBOLS

b	number of blades
R	blade radius, feet
r	radial distance to blade element, feet
c	blade-section chord, feet
t	blade thickness
c_e	equivalent blade chord, feet $\left(\int_0^R cr^2dr \Big/ \int_0^R r^2dr\right)$
σ	rotor solidity $\left(\frac{bc_e}{\pi R}\right)$
ρ	mass density of air, slugs per cubic foot

Ω	angular velocity of rotor, radians per second
β	blade flapping angle
α_r	blade-element angle of attack measured from line of zero lift, degrees
δ_3	angle in the plane of rotation between perpendicular to blade-span axis and flapping-hinge axis, positive when an increase in flapping produces a decrease in blade pitch, degrees
T	rotor thrust, pounds
Q	rotor-shaft torque, pound-feet
C_T	rotor-thrust coefficient $\left(\frac{T}{\pi R^2 \rho (\Omega R)^2}\right)$
C_Q	rotor-shaft-torque coefficient $\left(\frac{Q}{\pi R^3 \rho (\Omega R)^2}\right)$
v	induced-inflow velocity at rotor, feet per second
v_{hov}	induced-inflow velocity at rotor in hovering, feet per second
V	true airspeed of helicopter along flight path (used herein as the true wind velocity relative to rotor), feet per second
K	blade torsional stiffness, pound-inches per radian

APPARATUS AND TESTS

Description of the Rotor

General rotor configuration.- The rotor tested was designed to operate at a rotor speed of 220 to 240 rpm and a thrust of 1350 pounds. The configuration is a two-blade teetering system with blades fastened rigidly at the hub and twisted outboard by auxiliary airfoils, called flaps, to obtain pitch control. A general view of the rotor installation on the test tower is presented as figure 1.

The rotor hub is attached to the shaft by means of a single, horizontal, tapered pin which allows the blades to teeter. The hub is alined with this pin so as to give a delta effect at the zero-lag-angle

position by rotating the center line of the drag hinge forward 30° from the normal to the horizontal axis. Lag-angle motion of the blades in the plane of rotation is provided for by drag hinges located 7 inches from the center line of the shaft. This motion is partly restricted by friction dampers which are preset to a value of 90 pounds friction force before slippage occurs. These dampers provide interblade damping. Figure 2 shows details of the rotor-hub installation on the helicopter tower.

Blades.- The blades are made of laminated Sitka spruce and vary from a rectangular cross section at the blade root, through a modified Clark Y airfoil section, to a Clark Y airfoil section outboard of the 22-percent station. General views of the blades are shown in figure 3. The blades are untwisted but have an initial pitch angle of 2.8° and a preset coning angle of 6°. The rotor radius is 19 feet. The blades have a 6-inch chord from the 22-percent to the 64-percent station with a straight taper up to an 8-inch chord at the 74-percent station. The outer 26 percent is rectangular in plan form and has an 8-inch chord. The thickness varies from 37.5 percent of the chord at the 22-percent station to 9 percent at the 95-percent station. Blade dimensions are given in figure 4. Inasmuch as the usual position of the center of gravity of a solid Clark Y airfoil section is near 44 percent chord, the center of gravity of the outboard 26 percent of the blade was moved forward by weights placed in the nose of the flap bracket and by two streamline brass counterweights attached to the leading edge of the blades at the 87-percent- and 95-percent-radius stations. In this outboard section, a series of chordwise lightening holes was cut in the rearward section of the blade to aid in moving the center of gravity forward. This section was then covered with $\frac{1}{32}$-inch birch plywood.

Flap.- Pitch control of the rotor is obtained by twisting the blades by means of an aerodynamic servomechanism which is an externally mounted auxiliary airfoil called a flap. The flap is an NACA 0010.5 airfoil having a span of 35.35 inches, an area of 125 square inches, and an aspect ratio of 10. The maximum chord at the center of the span is 5.58 inches. The flap, like the blade proper, is constructed of laminated Sitka spruce.

The attachment of the flap to the blade itself is accomplished with an aluminum bracket fastened to the blade at the 75-percent station. The flap pivots on a hinge located at the trailing edge of this bracket and is mounted 0.82 inch outboard of the center of the flap span and 4 inches back of the blade trailing edge.

The flap-actuating mechanism is a mechanical-linkage system of bell cranks and push-pull rods which come up inside the hollow rotor drive shaft to the rotor head, pass inside the leading edge of the blade

to the flap bracket, then chordwise under the flap bracket to the flap hinge, which is bolted to the lower surface of the flap. A decrease in the angle of attack of the flap causes an increase in the blade-pitch angle, whereas an increase in the flap angle results in a decrease in blade-pitch angle, nose-up flap deflections being considered positive.

Summary of rotor properties.- A summary of rotor properties is as follows:

Rotor-blade characteristics:	
Blade radius, feet	19
Blade twist (no flap pitch applied), degrees	0
Preset blade-pitch angle, degrees	2.8
Preset coning angle, degrees	6
Solidity (blade and flap), σ	0.023
Blade area (one blade), square feet	10.22
Blade section:	
Root to 22-percent radius	Modified Clark Y
22-percent radius to tip	Clark Y
Blade weight (one blade, including flap, flap bracket, and counterweights), pounds	54.5
Drag-hinge location from center of rotation, inches	7
Blade center-of-gravity location from drag hinge, inches	74.4
Blade moment of inertia (one blade, including flap, flap bracket, and counterweights) about drag hinge, pound-inch-seconds^2	1545
Offset of center line of flapping hinge from center of rotation, inches	0
Available rotor-tilt range, degrees	± 7
Flapping-hinge delta effect, δ_3, degrees	30
Natural torsional frequency, cycles per second	6.95
Natural bending frequency, cycles per second	1.1
Torsional stiffness at 0.75 radius, K, pound-inches per radian	1560
Flap characteristics:	
Span, inches	35.35
Area, square inches	125
Airfoil section	NACA 0010.5
Aspect ratio	10
Pivot-point location outboard from center of flap span, inches	0.82
Available flap angle range, degrees	15

Testing Methods

A general description of the tower and of most of the methods of measuring various quantities is given in reference 1. The quantities

measured during the tests were rotor angular velocity, rotor thrust, rotor-shaft torque, blade-pitch angle at 75 percent radius, tab-pitch angle (in relation to blade), blade flapping angle, blade drag angle, wind velocity, and induced velocities in the wake beneath the rotor. All data were measured with a recording oscillograph. The blade-pitch angle was obtained from electric torsion strain gages mounted on the blade, as well as with a camera mounted on top of the rotor shaft. The induced velocities were measured with small calibrated windmill anemometers located on a boom beneath the rotor disk.

Hovering-performance tests were made during dead-calm wind conditions. The performance of the rotor over a range of wind velocities from 0 to 30 miles per hour was also obtained. Wake velocities were obtained in hovering by taking a survey of the rotor wake at a thrust of 1350 pounds and a rotor speed of 240 rpm. Control effectiveness, or the magnitude of the collective blade-pitch change for a given collective flap-pitch change as well as the magnitude of the rotor tilt per degree of cyclic flap pitch, was found for various rotor speeds. Transient response of the rotor to collective-pitch increase was measured by displacing the collective control at various rates. A spring was used to obtain the very rapid collective-pitch changes. The same procedures were used in displacing the cyclic-pitch control to obtain the transient response to cyclic-pitch increase. The time lag between tab movement and main blade-pitch movement was also measured for various rotor speeds.

RESULTS AND DISCUSSION

Hovering Performance

Figure 5 presents the measurements of blade and flap pitch angles under hovering conditions for a rotor speed of 220 rpm and over a range of rotor thrust from 0 to 1500 pounds. The blade angle is that measured from camera records of the 75-percent-radius station. Zero pitch angle is taken as that position at which the straight bottom surface of the Clark Y airfoil is horizontal. The flap angles are referenced to the blade angle; zero pitch angle of the flap is taken as that position at which the chord line of the flap is parallel to the bottom surface of the blade.

The results of the hovering-performance measurements are presented in figure 6 as the variation of torque coefficient with thrust coefficient. Calculations made from these data indicate that 60.8 horsepower is required to produce 1350 pounds thrust at the lower operating speed of 220 rpm as compared to 65.1 horsepower at the higher operating speed of 240 rpm. An increase in C_Q, as C_T is decreased, is shown in the low range of C_T. This increase may be considered as arising from two

sources. First, to produce zero thrust, the outer portion of the blade must operate at a negative pitch angle to counteract the lift produced by the inboard sections, which are fastened rigidly at the root and always have a positive pitch angle. Second, the flap must operate at a high positive pitch angle to produce a moment to twist the outer portion of the blade to the negative pitch angle. This increase in C_Q, however, is considered as not hampering the rotor operation since it occurs near the zero-thrust condition.

Inasmuch as the value of C_Q at zero thrust coefficient is not representative of the profile drag of this configuration and inasmuch as an attempt to obtain the profile drag of the blade experimentally by operating without the flap assembly was deemed inadvisable because of the probability of a low flutter speed, an estimate of the profile drag of the flap could be obtained only by comparing the performance of this configuration with the calculated performance of a conventional rotor with untwisted blades and the same diameter and solidity as the test rotor but without the flap bracket. Accordingly, the performance of such a rotor was calculated by the method presented in reference 2, by use of a drag curve representative of well-built plywood blades as given in reference 3. The performance curve is presented as a dashed line in figure 6. A comparison of the two curves would yield an estimate of the profile drag of the test rotor if its induced losses could be determined and compared with the calculated induced losses of the conventional rotor.

In order to determine the induced losses, a wake survey was conducted under hovering conditions at a thrust of 1350 pounds and a rotor speed of 240 rpm. The wake velocities were measured at a distance of 22 inches (about 10-percent radius) below the center line of the rotor hub. The results are shown in figure 7. A comparison of these results with unpublished NACA experimental data on twisted and straight conventionally controlled rotor blades indicates that the hovering-induced losses of this rotor are comparable to those of a conventional rotor of the same diameter having approximately 3° to 5° of lineal washout.

Since the wake-velocity survey indicates that the induced losses of the test rotor and the conventional rotor differ by approximately 1 percent, the remaining difference in performance between the calculated curve and the experimental curve is expected to be due to profile drag. The power required for the test rotor to hover at a thrust of 1350 pounds and a rotor speed of 240 rpm is approximately 6.5 percent greater than that estimated for a conventional rotor of the same diameter and solidity. It is believed that most of this difference is caused by the flap servo-mechanism. Some reduction of the flap profile drag could be expected with more streamlining of the flap controls and assembly. The total improvement in performance in the operating range, however, will probably not be greater than 3 percent of total power.

Low-Speed Flight Performance

A limited amount of performance data were obtained with the rotor over a range of wind velocities from 0 to 30 miles per hour. The ratio of the induced power in hovering to the induced power in forward flight, as represented by the ratios $\frac{v}{v_{hov}}$ and $\frac{V}{v_{hov}}$ was obtained experimentally by using the method of reference 1. The results are shown in figure 8 and are in good agreement with the theoretical and experimental results for rotors with conventionally actuated pitch.

The measurements of the total power required for sustentation at a rotor thrust of 1350 pounds and a rotor speed of 220 rpm are presented in figure 9 as a conventional plot of horsepower against wind velocity. The large decrease in power with increasing wind velocity is accentuated by the low hovering-induced velocity necessary for 1350 pounds thrust, which gives an unusually low disk loading. This disk loading was chosen to give normal values of mean lift coefficient with the low-solidity rotor used.

Control Effectiveness

The magnitude of the blade-pitch change for a given pitch change of the flap was found to increase with increasing rotor speed and to depend on whether the flap pitch was applied cyclicly or collectively. Figure 10 shows the results for both cases. The top solid curve gives the amount of rotor tilt per degree of flap cyclic pitch for the various rotor speeds. For a rotor speed of 240 rpm, a 1° change in the flap cyclic pitch causes a change in the rotor tilt of approximately 2°.

This experimental cyclic control effectiveness includes the effect of a 30° δ_3 hinge angle, whereas the more basic case would be 0° δ_3 hinge angle. Inasmuch as no experimental data are available for a similar rotor having 0° δ_3 hinge angle, it is desirable, therefore, to estimate the effect of the 30° δ_3 hinge angle. An approximate analysis indicates the following effects: (1) the control effectiveness is reduced by using the 30° δ_3 hinge and (2) the phase angle in azimuth between the position of maximum cyclic blade pitch and the maximum flapping is reduced by using the 30° δ_3 hinge.

In regard to effect (1), the amount that the δ_3 hinge would normally reduce the control effectiveness is partly nullified because the pitch change effected at the blade root by the δ_3 hinge is not wholly transferred to the outer portion of the blade because of its low torsional stiffness. The increase in torsional stiffness due to centrifugal force and blade inertia was considered; however, the analysis indicated that it had no significant effect on the amplitude of the cyclic-

control effectiveness. The effective δ_3 hinge angle then is proportional to the ratio of the effective torsional stiffness of the blade to the moment produced by the flap which for this case is approximately 1/4. The simplified equation $\Delta\alpha_r = \beta \tan \delta_3$, which represents the blade-pitch change due to flapping with the δ_3 hinge for torsionally rigid blades, would therefore be modified for this configuration to be $\Delta\alpha_r = \frac{1}{4}\beta \tan \delta_3$. Solving this expression for an equivalent effective δ_3 hinge angle yields a value of approximately 8°. This indicates that $\Delta\alpha_r$ is approximately $\beta \tan 8^\circ$ and that the control effectiveness is reduced approximately 3 percent by the δ_3 angle used.

Effect (2) is experimentally verified by the test data which show a phase angle of approximately 70° between the positions of maximum cyclic blade pitch and maximum flapping which is 20° less than would be expected with 0° δ_3 hinge angle. The data also show a phase-angle lag of approximately 30° in the cyclic-blade-pitch response to the cyclic flap-pitch motion. The total phase angle between the cyclic flap motion and the resulting flapping is thus approximately 100°, which is 10° higher than results expected with a conventional rotor with 0° δ_3 hinge angle.

The lower dashed curve of figure 10 shows the amount of collective blade-pitch change per degree of collective flap pitch. For a rotor speed of 240 rpm, a 1° change in the collective flap pitch results in a 1.6° change of blade pitch, as measured at the 75-percent-radius station. The variation of control effectiveness with resultant velocity at the flap may be undesirable with respect to satisfactory flying qualities since it could be a source of vibration at higher tip-speed ratios.

The measurements show that with an available flap-angle range of 15°, sufficient control was available at normal rotor speeds to give a rotor tilt range of $\pm 7^\circ$ with respect to the shaft and a thrust range of 0 to 1500 pounds.

Transient Response to Collective Pitch

Several tests were made to determine the transient response of the rotor to various rates of collective flap-pitch change. The time history of a typical record is shown as figure 11. The rate of flap-pitch increase is approximately 20° per second, which is thought to be the maximum rate at which a pilot could move the controls. The slight increase in the flap pitch after it reached its first maximum value is due to the mechanical coupling between the lag angle and the flap pitch. As the blade is displaced rearward or forward from the no-lag position, the flap pitch is increased slightly; however, this is a secondary effect and does not materially affect the record. The thrust responds approximately 0.08 second after the flap-pitch change is initiated and reaches

a maximum value about 0.05 second after the flap pitch has reached its maximum value. The thrust is seen to overshoot momentarily, oscillate for a few tenths of a second, and attain its final value approximately 1 second after the flap-pitch change is completed.

The horsepower input shows a temporary decrease as the thrust increases, followed by an increase and then a mild rising oscillation. The power is still increasing when the thrust and pitch angle have come to a steady value. The final value of the power is reached at 3 to 4 seconds after the flap pitch reaches its final value, probably because the kinetic energy stored in the blades is fed back into the system as the rotor slows down. As a result, the drive torque does not increase as rapidly as the thrust.

The time history of a much faster increase in collective flap pitch is shown in figure 12. In this run, the flap pitch was actuated by a strong spring at a rate almost corresponding to an instantaneous step deflection. The very rapid rate was used to determine the response and the stability of the configuration to an effective step change in the flap pitch. The flap pitch oscillates for two cycles after it reaches a maximum value. This oscillation is caused by vibration of the control stop and the play in the various linkages, but it is thought that the results are not affected. The thrust shows a response approximately 0.05 second after the flap-pitch change is initiated, reaches a maximum value about 0.07 second after the flap pitch reaches a maximum, oscillates violently for several cycles, and comes to its steady-state condition in approximately 1 second after the flap-pitch change is completed. The blade pitch responds approximately 0.03 second after the flap-pitch change is initiated and reaches its maximum value approximately 0.03 second after the flap-pitch change is completed. In general, it was found that the time lag between flap motion and the blade-pitch response is approximately 0.02 to 0.03 second. The response of the test rotor following the blade-pitch response is similar to that of a rotor with conventionally actuated pitch changes. The blade pitch also oscillates violently for several cycles and attains its steady-state condition approximately 1 second after the flap-pitch change is accomplished. The blade-pitching frequency is probably a coupling of both the bending and the torsional frequencies. The rotor-shaft speed shows a steady decrease because the test was run at a constant throttle setting. The horsepower input shows a temporary decrease when the flap pitch is first changed, followed by a slow increase to the final value 4 to 5 seconds after the flap pitch reaches a maximum value. The damping of the blade pitch and thrust oscillations after the very rapid change of the flap pitch indicates that the blade-flap configuration is stable for this type of disturbance.

Transient Response to Cyclic Pitch

Several tests were made to determine the transient response to various rates of cyclic flap-pitch change. The time history of a typical slow-rate run under hovering conditions at a rotor thrust of 1350 pounds and a rotor speed of 225 rpm is shown in figure 13. The cyclic-pitch rate of change is approximately 3.33° per second. The rotor tilt responds approximately 0.06 second after the cyclic flap-pitch change was initiated and reached a maximum value approximately 0.2 second after the cyclic flap pitch reached a maximum value. The time lag between the rotor tilt and applied cyclic flap pitch, measured at the zero axis, is approximately 0.11 second; 0.08 second is the time lag obtained for conventional rotors and the remaining time may be regarded as the time lag between the flap motion and blade-pitch response. A similar delay is seen when the flap pitch and consequently rotor tilt is returned to its original position.

A time history of a very rapid change in cyclic flap pitch at a rotor thrust of 1350 pounds and a rotor speed of 235 rpm is presented in figure 14. The very rapid rate was obtained by actuating the cyclic control with a strong spring and was used to establish the response and stability of the configuration to an effective step change in the cyclic flap pitch. The flap pitch reaches its maximum value and shows a slight oscillation due to vibration in the control stop, but this effect should not materially affect the results. The rotor tilt shows a response to the flap pitch approximately 0.06 second after the flap-pitch change is initiated and reaches full tilt approximately 0.27 second after the flap-pitch change is completed. The cyclic blade pitch, as indicated by the blade torsion gages, shows a response approximately 0.02 second after the flap pitch is changed and builds up to full deflection in approximately 0.15 second. The rate of change of blade pitch shown in figure 14 was the maximum that could be obtained inasmuch as the rate is limited by the natural frequency of the blade-flap configuration.

The time histories shown in figure 13 and 14 indicate that the response to the flap cyclic controls is highly damped and that the blade-flap configuration is stable with respect to flap cyclic changes. They also indicate that the time lag is well within the accepted requirement of control response as given in reference 4.

CONCLUSIONS

The performance and control characteristics of an aerodynamic-servocontrolled rotor system were experimentally determined on the

Langley helicopter tower. On the basis of these tests, the following conclusions may be drawn:

1. The power required for the test rotor to hover at a thrust of 1350 pounds and a rotor speed of 240 rpm is approximately 6.5 percent greater than that estimated for a conventional rotor of the same diameter and solidity. It is believed that most of this difference is caused by the flap servomechanism.

2. The reduction with increasing wind velocity in total power required for sustentation of the configuration tested at the normal operating rotor thrust was found to be similar to the theoretical and experimental results for rotors with conventionally actuated pitch as reported in NACA TN 1698.

3. The control effectiveness was found to vary with rotor speed. With an available flap-angle range of 15°, sufficient control was available at normal rotor speeds to give a rotor tilt range of ±7° with respect to the shaft and a thrust range of 0 to 1500 pounds.

4. The blade-flap configuration appears stable with respect to either collective- or cyclic-pitch changes.

5. The time lag between flap motion and the blade-pitch response is approximately 0.02 to 0.03 second. The response of the test rotor, following the blade-pitch response, is similar to that of a rotor with conventionally actuated pitch.

6. The over-all characteristics of the rotor as determined from these tests indicate that satisfactory performance and control characteristics can be obtained by using an aerodynamic type of servocontrol.

Langley Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Air Force Base, Va., October 14, 1949

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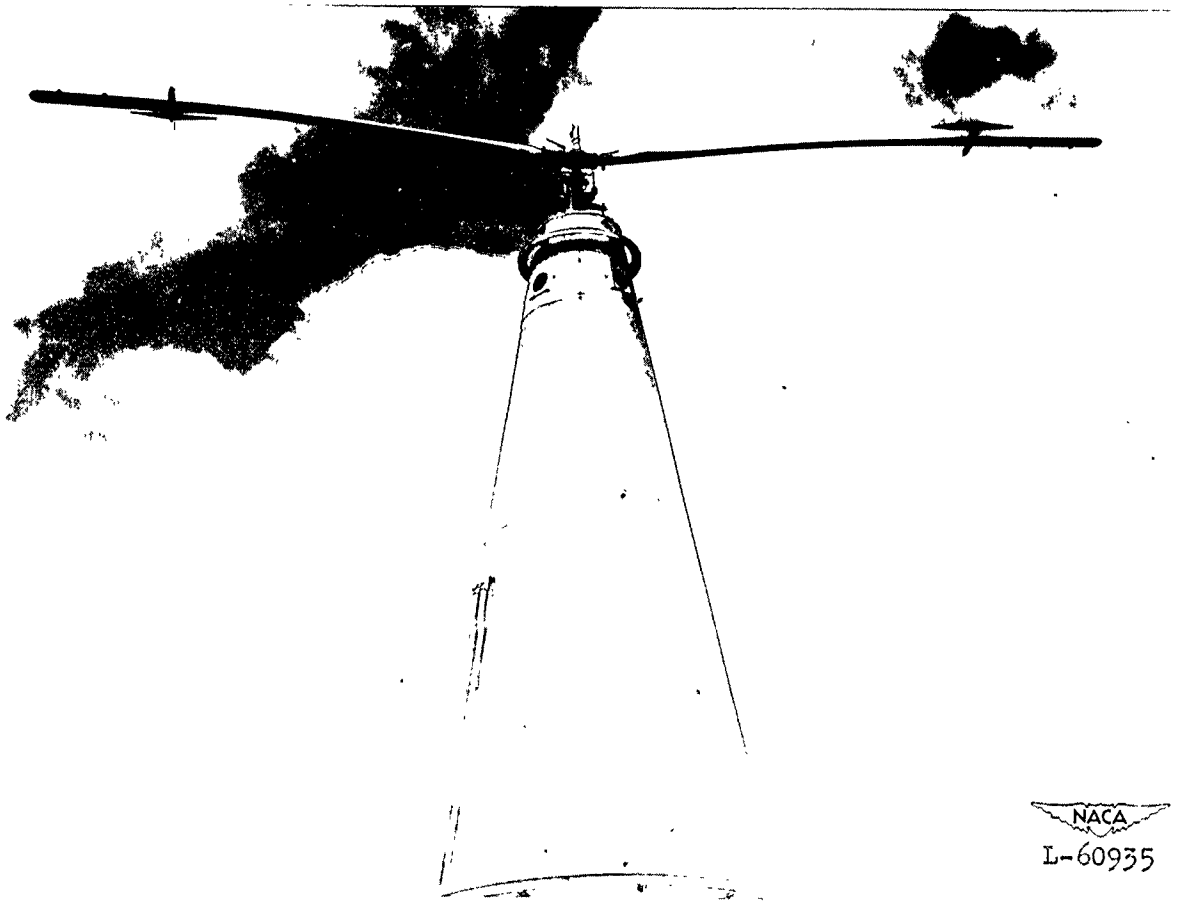


Figure 1.- General view of rotor installation on the Langley helicopter tower.

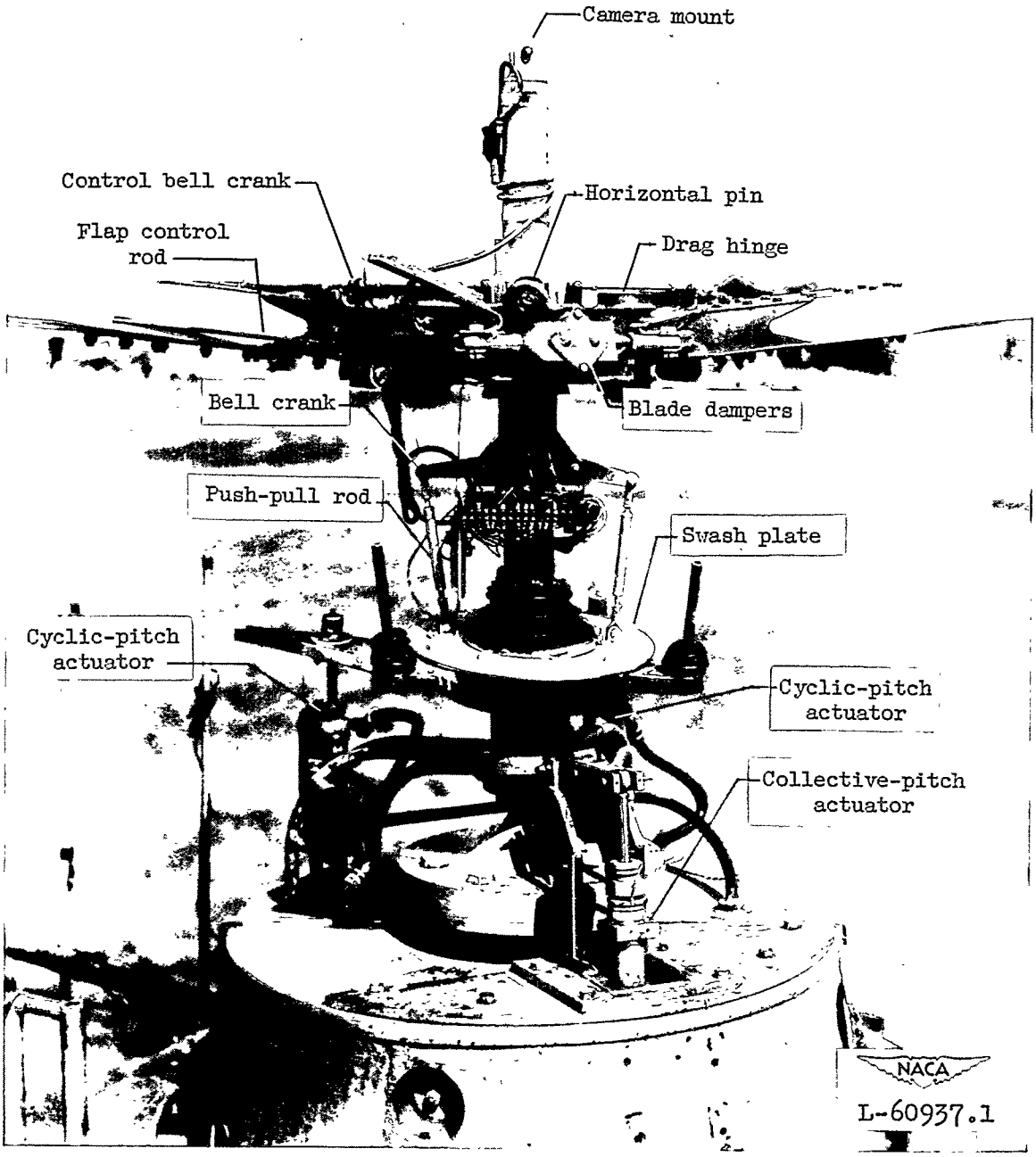
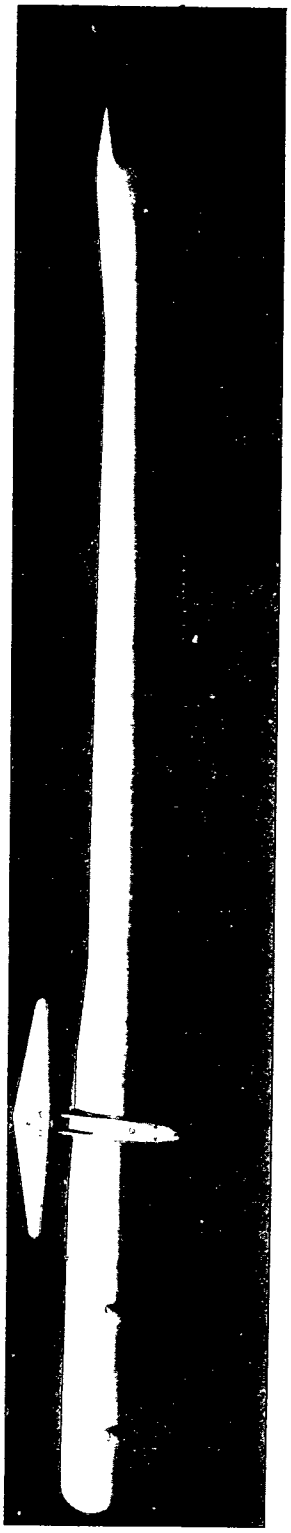
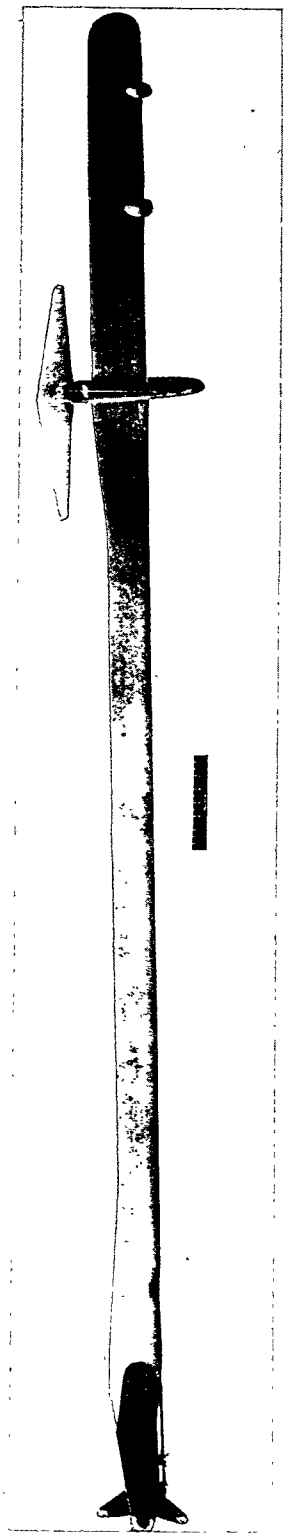


Figure 2.- Details of rotor-hub installation on the Langley helicopter tower.



(a) Top view.



(b) Bottom view.

NACA L-62129

Figure 3.- Plan-form views of test rotor blade.

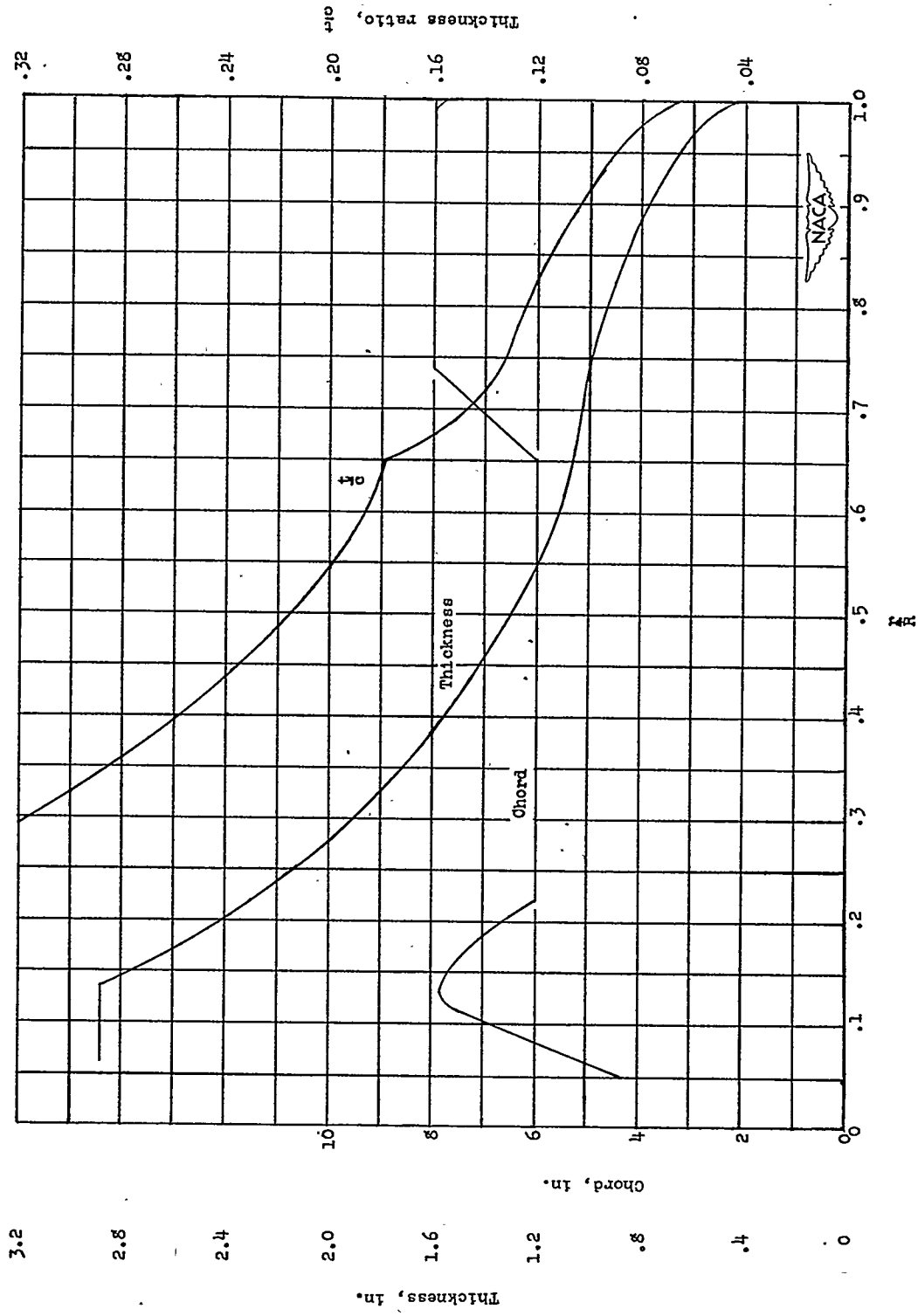


Figure 4.- Rotor-blade dimensions.

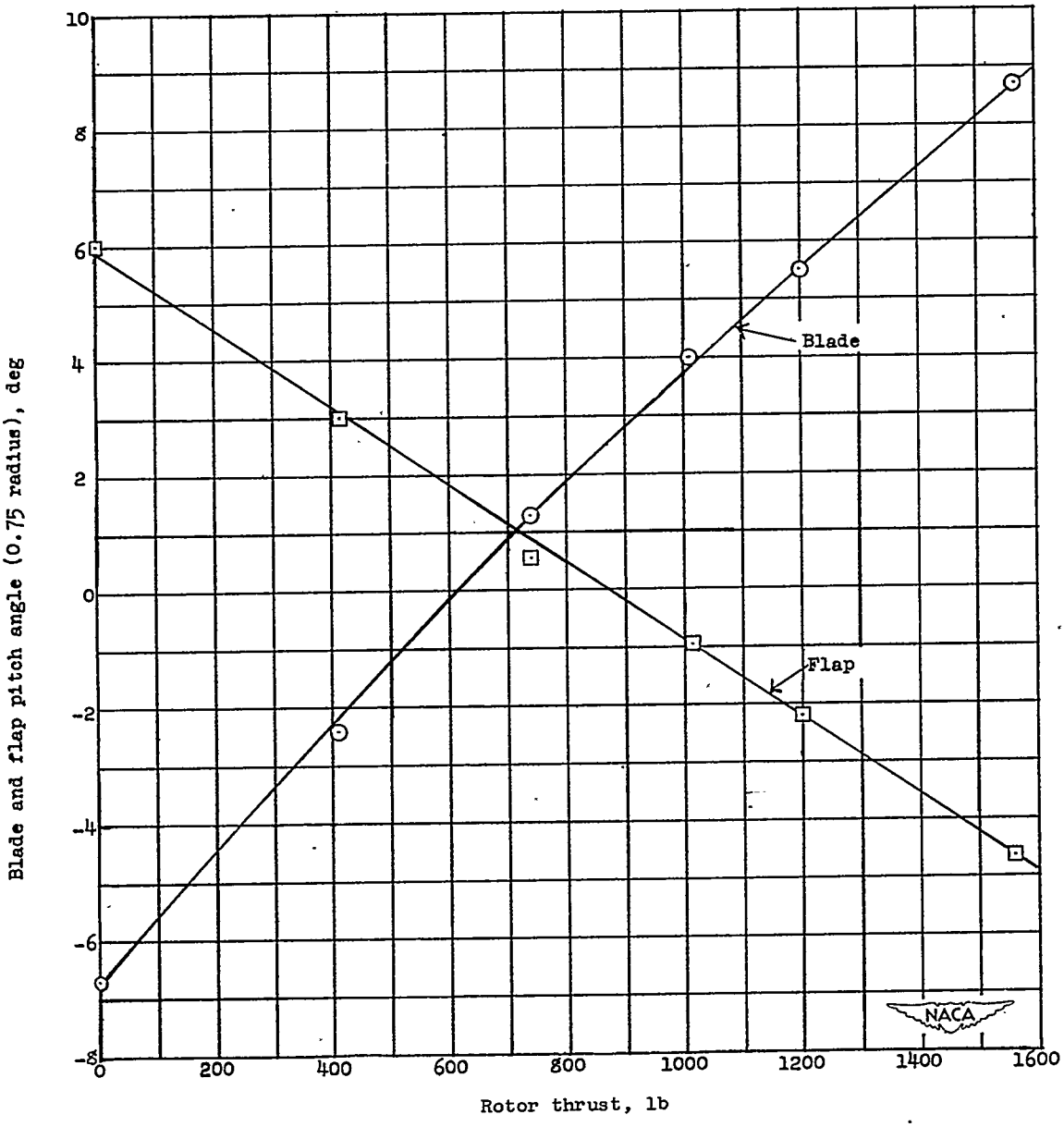


Figure 5.- Blade and flap pitch angles at hovering conditions and rotor speed of 220 rpm.

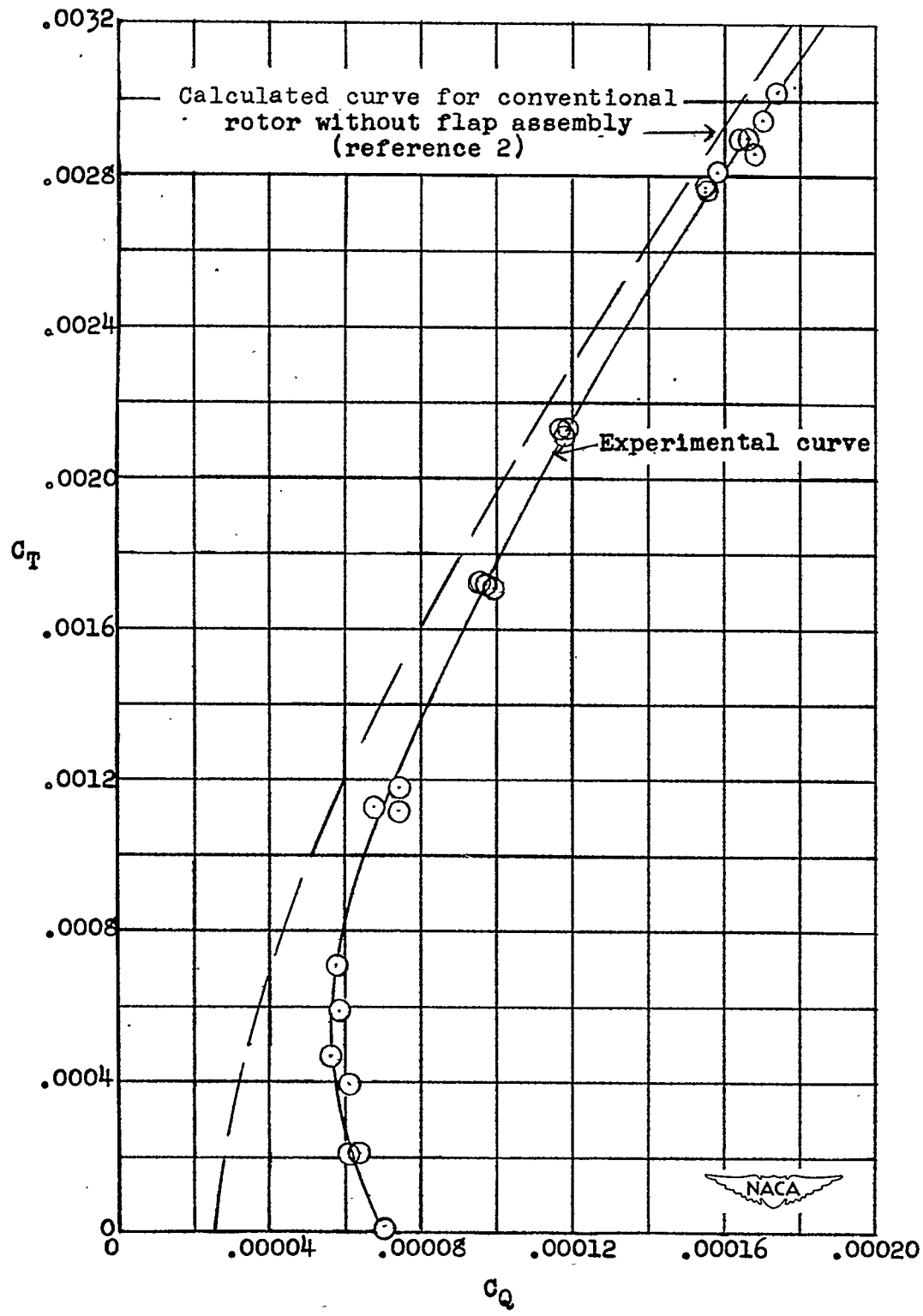


Figure 6.- Torque and thrust coefficients at hovering conditions.

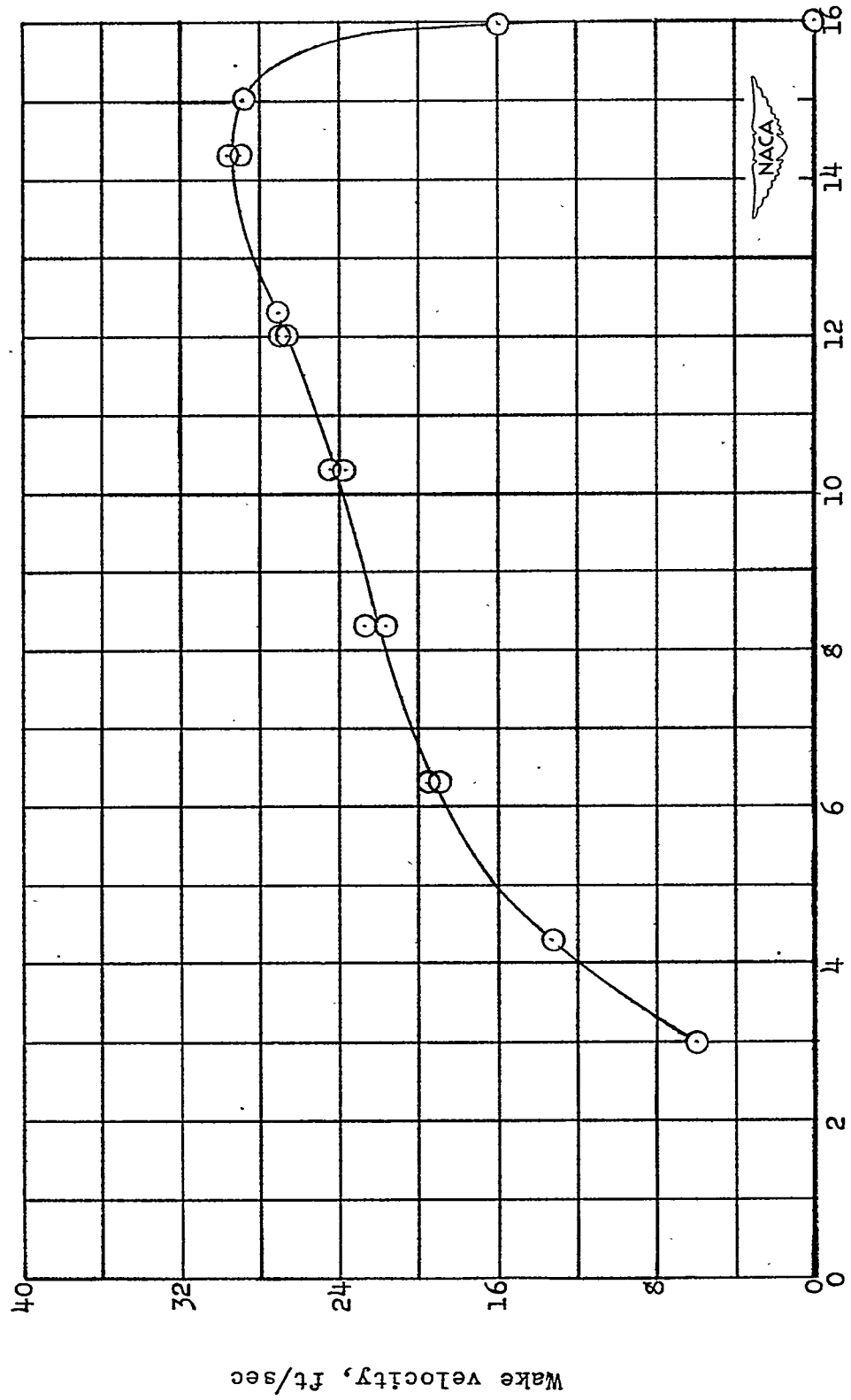


Figure 7.- Survey of wake velocities at hovering conditions and rotor thrust of 1350 pounds as measured 22 inches (0.10 R) below center line of rotor hub.

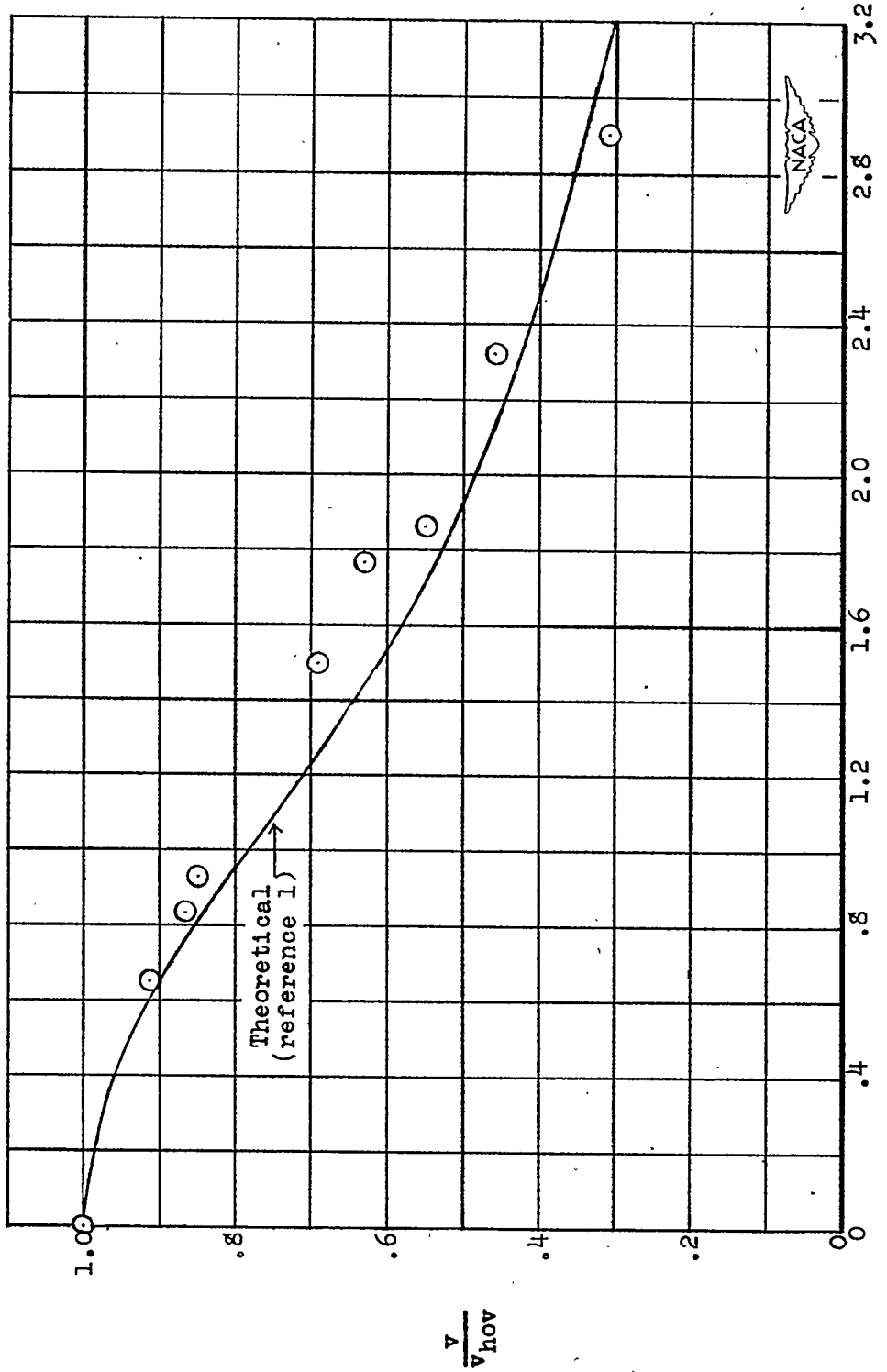


Figure 8.- Comparison of theoretical and experimental values of induced-velocity parameter $\frac{v}{v_{hov}}$ against wind-velocity parameter $\frac{V}{v_{hov}}$.

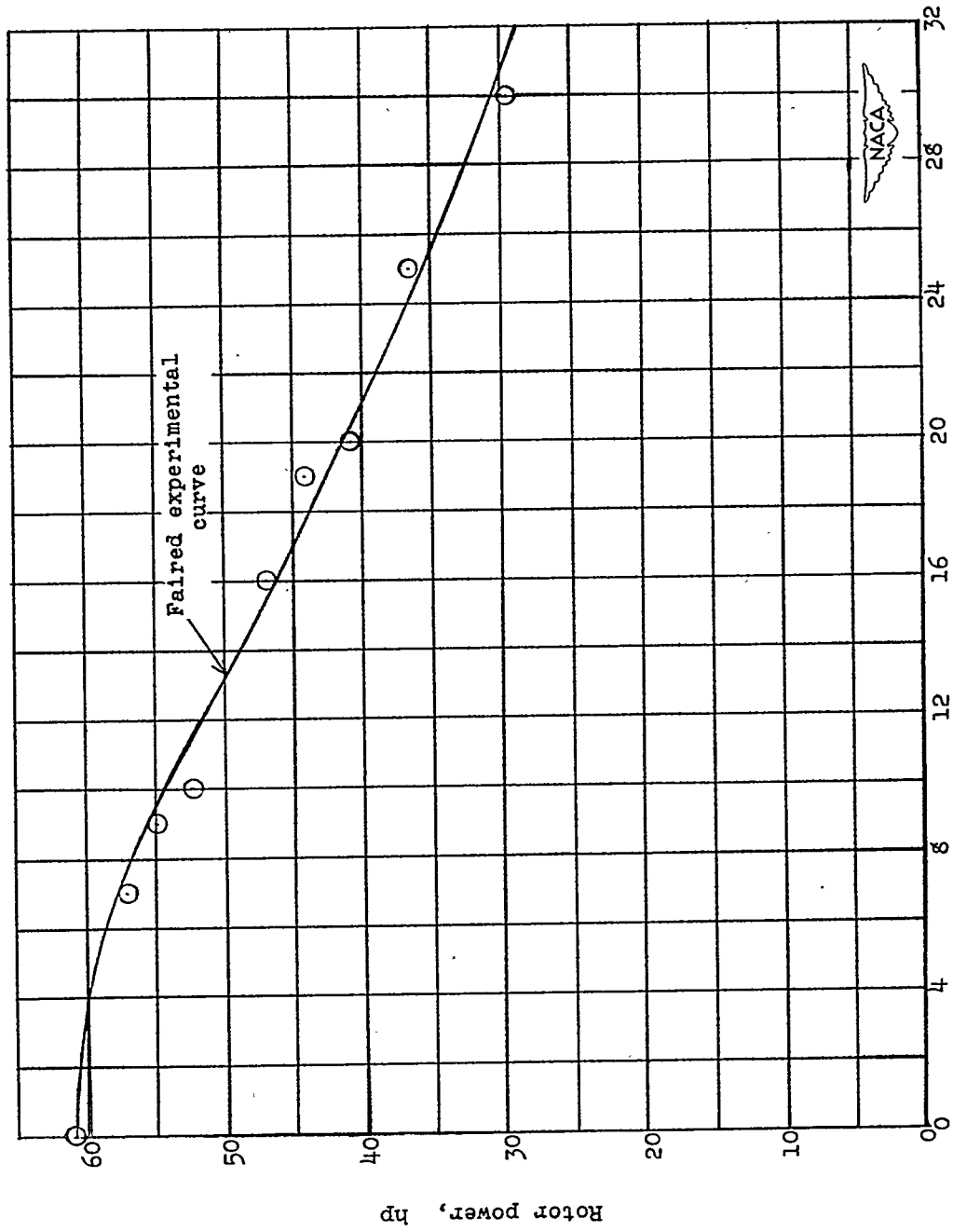


Figure 9.- Effect of wind velocity on the power required for sustentation at a rotor thrust of 1350 pounds and a rotor speed of 220 rpm at sea level.

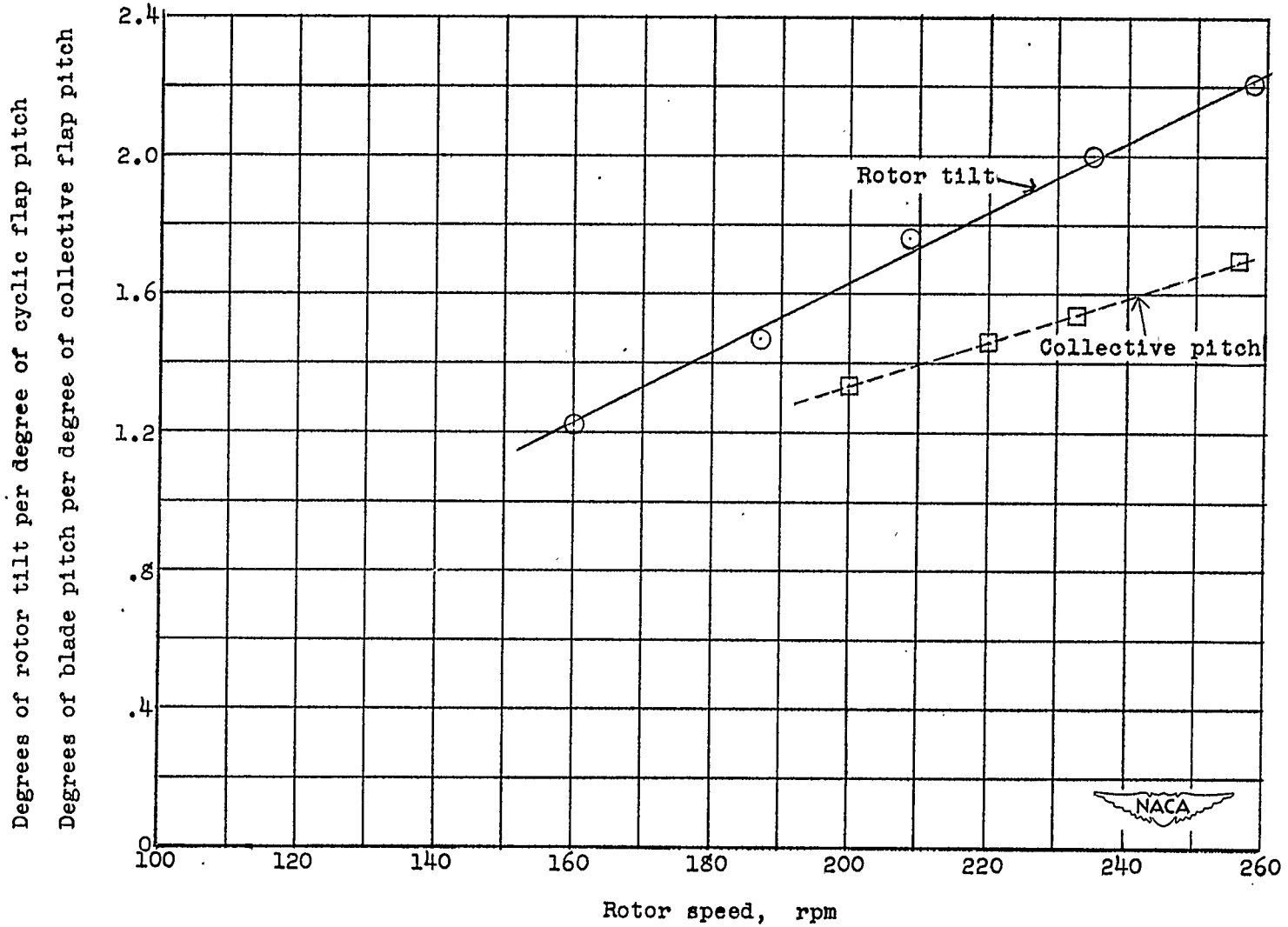


Figure 10.- Control effectiveness for both collective and cyclic flap pitch at various rotor speeds.

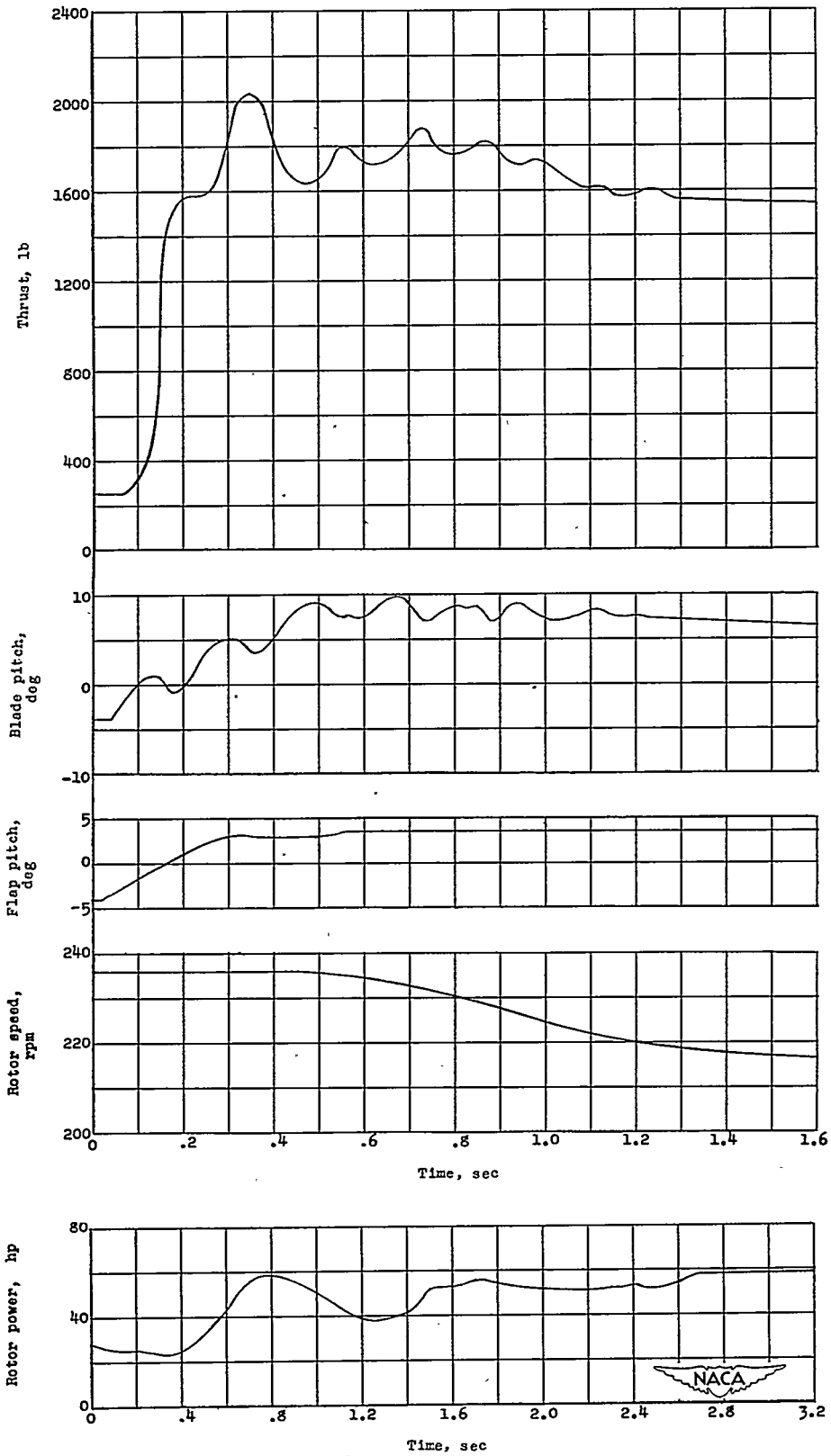


Figure 11.- Time history of rapid collective-pitch increase.

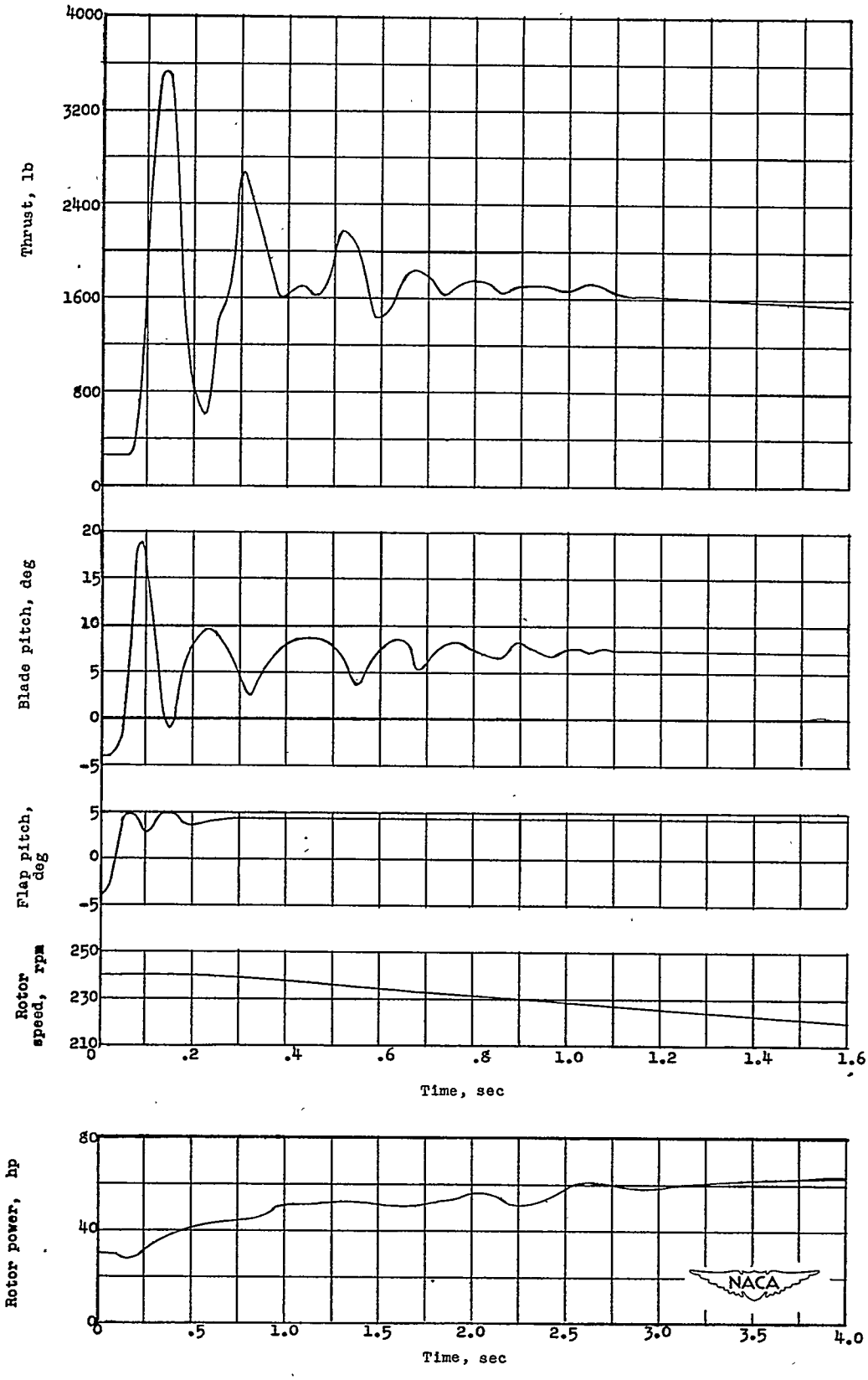


Figure 12.- Time history of very rapid collective pitch increase.

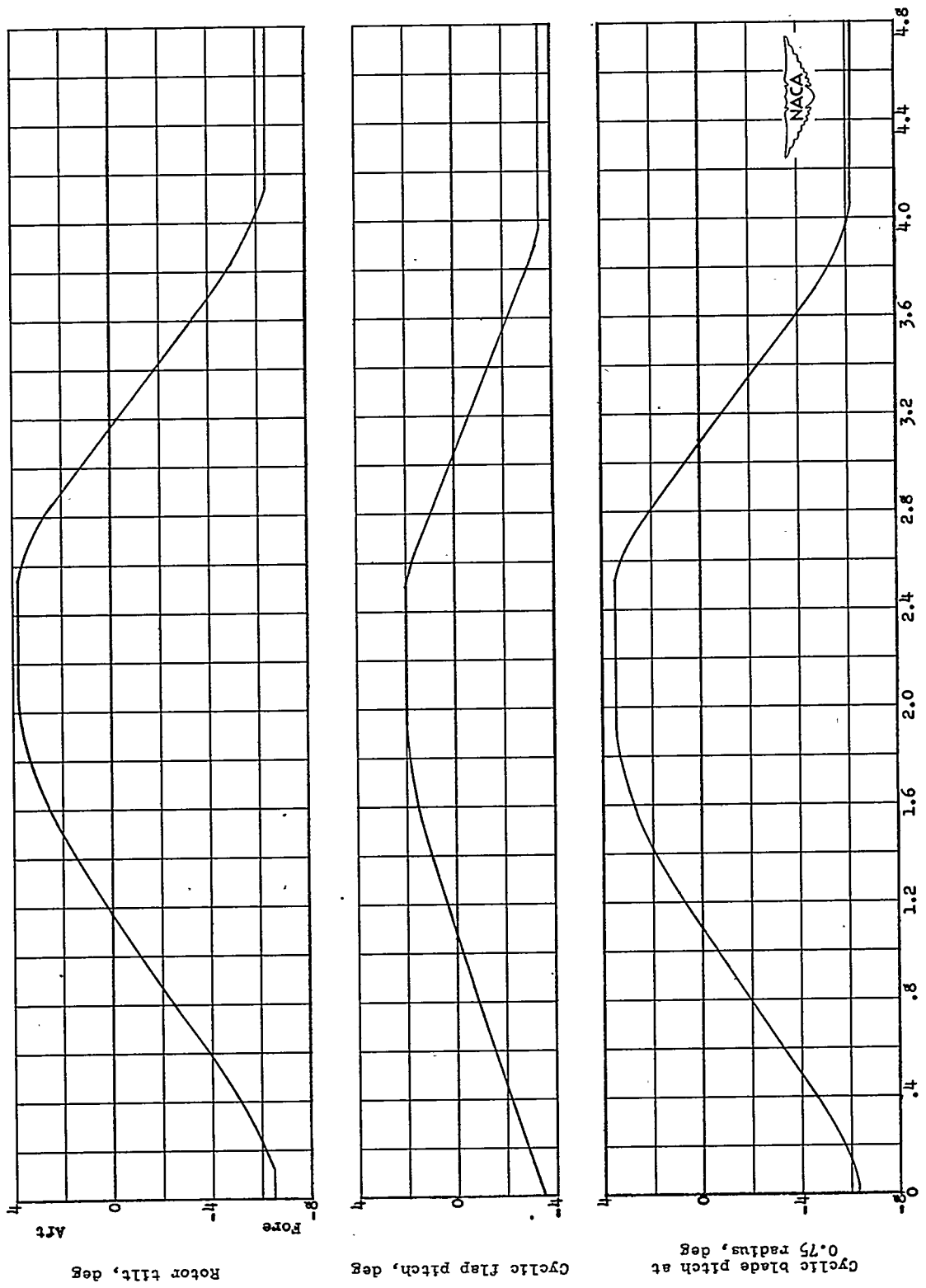


Figure 13.- Time history of slow cyclic flap-pitch change under hovering conditions at rotor thrust of 1350 pounds and rotor speed of 225 rpm.

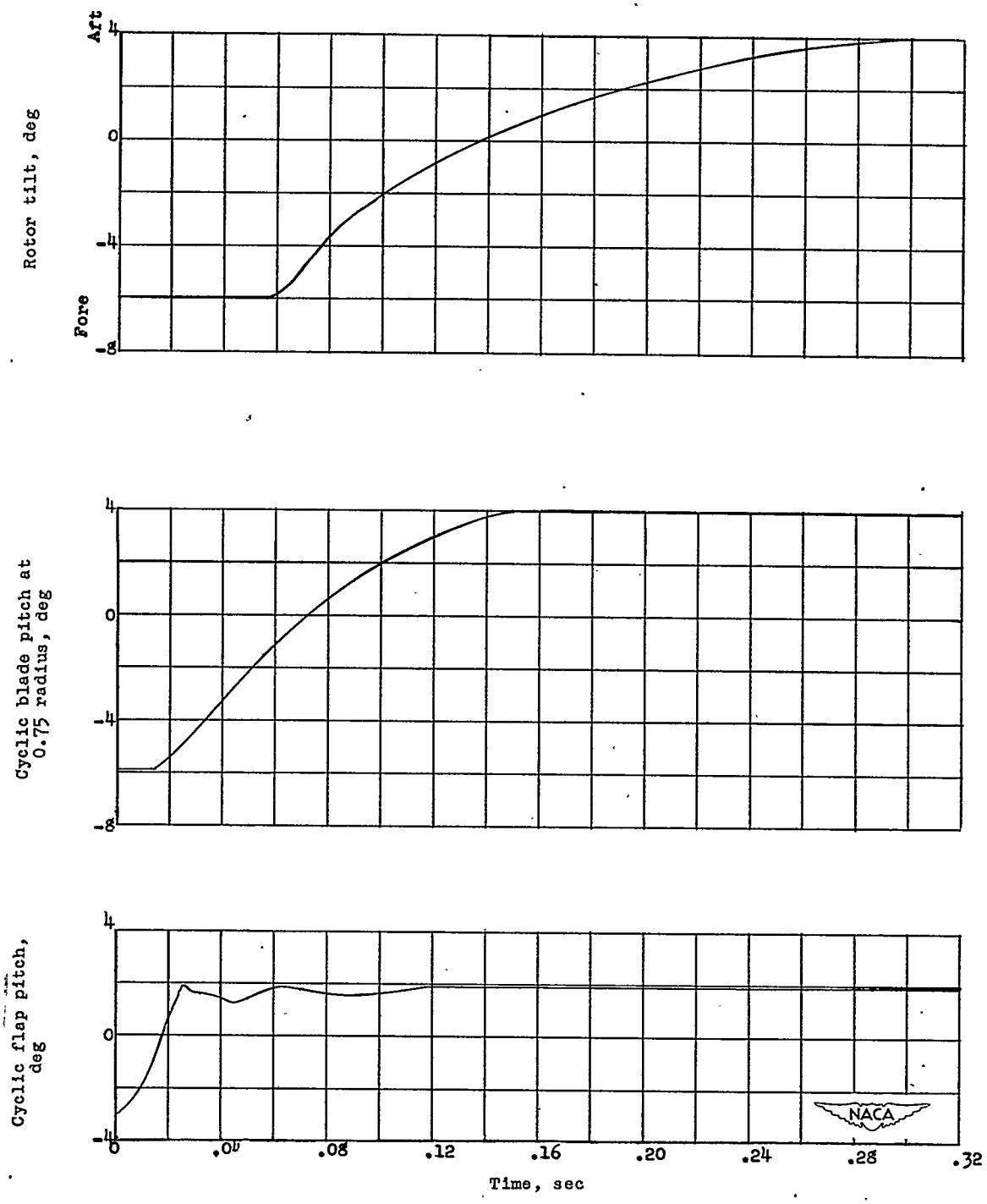


Figure 14.- Time history of very rapid cyclic flap-pitch change at rotor thrust of 1350 pounds and rotor speed of 235 rpm.

NACA-Langley - 5-12-50 - 925